Propulsion for Interplanetary Space Missions

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Most of the available interplanetary mission studies have evaluated either the heliocentric velocity requirements for transfer between planetary orbits, or the performance of rockets departing and entering planetocentric orbits. The purpose of this paper is to present a method of relating the interplanetary transfer velocity requirements to a space vehicle and its propulsion system performance.

The method developed provides a means of analyzing the major propulsion phases of interplanetary missions initiating and terminating in circular planetocentric orbits. This method provides the basis for a realistic, integrated analysis of propulsion system requirements for specific interplanetary missions, including the effect of depature date and transfer time on propulsion requirements. With this method, the effect of each mission and propulsion parameter can be examined with regard to overall vehicle performance.

Preliminary evaluations of interplanetary space vehicles are presented to indicate typical analyses of payload capabilities and propulsion requirements for space vehicles.

The paper is divided into three parts. The first presents the scope of the mission analysis and introduces the interplanetary mission parameters. The second develops the nomograph analysis method, and the third presents representative mission evaluations to indicate use of the nomograph method and offers typical results of space propulsion analyses.

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Fundamentals of Interplanetary Space Missions

Scope of Mission Analysis

Since relative positions of the planets at initiation of an interplanetary mission affect the propulsion requirements, a three-dimensional description of the solar system configuration has been considered necessary for a comprehensive analysis. The solar model used in this analysis assumes that the planets revolve about the sun in elliptical orbits. In this solar system model, the inclinations of the orbital planes as well as the eccentricity of the planetary orbits affect the spatial positions of the planets with respect to each other.

The interplanetary space missions (initiating from a circular geocentric orbit) analyzed in this paper are:

(1) Planetary probes.

(2) Planetary circular orbit establishment.

(3) Planetary circular orbit establishment and return to the earth.

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Fundamentals of interplanetary space missions—both scope and analysis—initiating from a circular geocentric orbit are discussed. Planetary probes, planetary circular orbit establishment and planetary circular orbit establishment with return to the earth are covered. Vehicles and propulsion systems are described and a nomograph method for analysis given.

Symbols

i_g = geocentric orbital inclination
i_H = heliocentric transfer plane inclination
T = launch date
τ = transfer time
V_h = earth hyperbolic excess velocity
V_a = planetary hyperbolic excess velocity
F = propulsion system thrust
I_s = rocket specific impulse
t = burning time
R = mass ratio
M = vehicle mass
$\dot{M}$ = mass flow rate
g_0 = earth surface gravitational constant
W = vehicle weight
V_p = local escape velocity
LO_2 = liquid oxygen
LH_2 = liquid hydrogen
λ_p = propellant fraction

The transfer trajectory for these missions consists of a portion of a heliocentric conic section mated by energy considerations at the boundaries to planetocentric hyperbolic trajectories. In these missions, propulsion maneuvers are required to (1) depart a planetocentric circular orbit and establish the planetocentric hyperbolic escape trajectory and (2) to change a planetocentric hyperbolic arrival trajectory to a circular orbit about a planet.

Generally, it is necessary to use digital computers in simulating the propulsion maneuvers (powered-flight trajectories) for evaluating each specific combination of mission and propulsion system parameters. The nomograph method was undertaken to provide a convenient analysis method (with comparable computer accuracy) which does not require

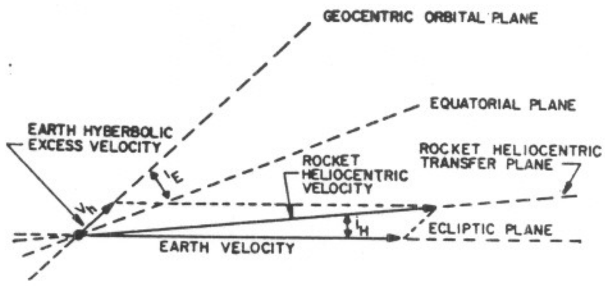

Fig. 1. Heliocentric plane change for planetary mission (executed at earth orbit departure).

new computer calculations for each change in propulsion parameters. The nomographs are based upon the results of computer-simulated trajectories for representative missions and space vehicles, and are valid for planar, tangential propulsion maneuvers.

Studies have shown that a thrust-aligned-with-velocity (tangential) maneuver (for orbit departure) yields near maximum performance for a propulsion system. Thus, a tangential program for thrust application has been selected. A planar propulsion maneuver can be defined as one in which the instantaneous thrust vector is coplanar with the instantaneous planetocentric position and velocity vectors throughout the propulsion phase.

A planar propulsion maneuver can be used to depart the Earth orbit and simultaneously establish the necessary heliocentric transfer plane, if the geocentric parking orbit plane has the proper orientation and inclination i_g to the equatorial plane. The vector addition of the rocket's hyperbolic excess velocity vector in the geocentric plane, with its mean heliocentric velocity vector (earth's velocity vector) in the ecliptic plane, determines the heliocentric transfer plane (Fig. 1). In the analysis method presented, it is assumed that the correct geocentric orbit

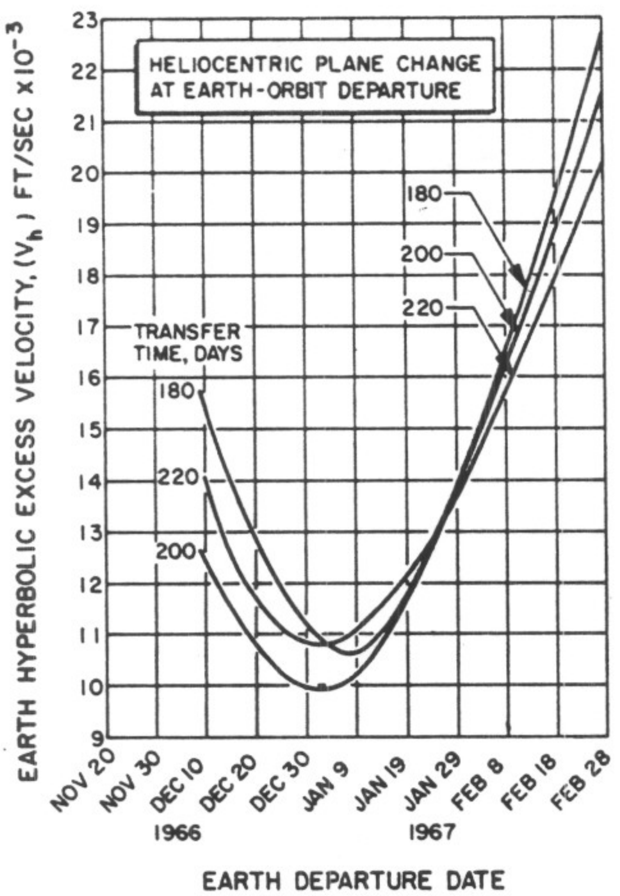

Fig. 2. Earth-Mars planetary mission.

inclination has been established prior to Earth orbit departure.

Selection of Mission Parameters

Breakwell[1, 2] has shown the dependency of the vehicle hyperbolic departure and arrival speeds (relative to a planet) upon the planetary positions at initiation and termination of the mission. His graphical presentations indicate the combinations of mission parameters—i.e., departure date and transfer time, that appear realistic when considering the propulsion system requirements.

A departure date T specifies the initial position of the rocket in the three-dimensional, heliocentric coordinate system. Although the rocket is initially in a geocentric orbit, its initial heliocentric position is assumed to be the same as the center of the earth. The transfer time τ combined with the departure date establishes the heliocentric rendezvous position of the rocket with the target planet. In heliocentric coordinates, the transfer trajectory must pass through the initial and rendezvous positions. Furthermore, the period of the heliocentric transfer conic must be such that the space vehicle will traverse the portion of the conic between departure and arrival in the indicated transfer time.

With the two spatial positions and the transfer time, an iterative computer program is used to determine the heliocentric conic and its boundry conditions. This conic is mated by energy considerations to planetocentric hyperbolic trajectories to complete the entire interplanetary transfer trajectory. These hyperbolic trajectories are characterized by planetocentric hyperbolic excess velocities—i.e., V_h for earth and V_a for planetary arrival or departure.

Some earth, Mars, and Venus hyperbolic excess velocities are presented as functions of earth orbit departure dates and transfer times in Figs. 2, 3, 4, and 5. These figures demonstrate how the mission parameters T and τ can be expressed as hyperbolic velocity requirements V_h and V_a which will be related to propulsion system and vehicle parameters.

Vehicle and Propulsion System Description

The powered flights associated with the hyperbolic trajectories constitute the major propulsion phases of an interplanetary space mission. These propulsion phases—i.e., earth orbit departure maneuver, planetary orbit establishment maneuver, and planetary orbit departure maneuver, are separated by extended periods of time. In addition, each of the maneuvers require large velocity changes. For these and other reasons, separate propulsion systems (stages) are used for each propulsion maneuver of a mission.

In an orbit about the earth, the rocket has a mean heliocentric velocity vector coincident with that of the earth. The propulsion system of the first stage supplies energy for the vehicle to leave the earth orbit and achieve the hyperbolic excess velocity (with respect to the earth) associated with the initial boundry conditions of the heliocentric conic.

At planet arrival, the heliocentric velocity vector of the vehicle will differ from that of the planet. The rocket will have hyperbolic excess velocity with respect to a planet-centered coordinate system. For planetary orbit establishment missions, a second stage propulsion system reduces the hyperbolic excess velocity of the rocket to affect capture by the planetary gravitational field.

The vehicle and propulsion system parameters to be used in the nomograph analysis method are F, I_s, W_0, t, and R.

These will be related to the two mission parameters which, previously, were related to departure date and transfer time: V_h and V_a.

Nomograph Method for Analysis of Interplanetary Missions

The nomograph relating the mission, vehicle, and propulsion system parameters is developed from rocket theory equations and computer-simulated trajectories in the manner described in the following paragraphs.

The instantaneous mass ratio for a rocket stage is defined as

$$R_t = M_0/M_t \qquad (1)$$

For a constant mass flow rate the instantaneous mass can be written

$$M_t = M_0 - \dot{M}t \tag{2}$$

Thus, the instantaneous mass ratio can be expressed as

$$R_t = M_0/(M_0 - \dot{M}t) \tag{3}$$

This mass ratio can also be

$$R_t = \frac{1}{1-(Ft/g_0M_0I_s)} \tag{4}$$

where g_0 is the earth surface gravitational acceleration. The product g_0M_0 is the initial earth surface weight W_0 of the vehicle, so Eq. 4 becomes

$$R_t = \frac{1}{1-(Ft/W_0I_s)} \tag{5}$$

For a propulsion maneuver characterized by constant thrust, and a burning time t, the burnout mass ratio R_{BO} is readily calculated from Eq. 5. As a result of the propulsion maneuver, a vehicle in space free of inverse square force fields would undergo a velocity change

$$\Delta V_I = I_s g_0 \ln R_{BO} \tag{6}$$

The burnout velocity of the vehicle would then be

$$V_{BO} = V_0 + I_s g_0 \ln R_{BO} \tag{7}$$

where V_0 is the initial velocity.

However, space mission propulsion maneuvers take place in inverse square force fields and, consequently, velocity losses V_L result. The velocity losses are an integral function of the burning time, the thrust orientation program and the planetocentric altitude of the vehicle which continuously changes during the propulsion phase.[3] The burnout velocity must be obtained by numerical integration of the equations of motion from time zero to time t. The computed burnout velocity may be expressed

$$V_{BO} = V_0 + I_s g \ln R_{BO} - V_L \tag{8}$$

which differs from Eq. 7 by the velocity losses term.

For space mission analysis, the burnout velocity for propulsion maneuvers in the vicinity of a planet and the escape velocity V_p at the burnout position are combined in the energy equation

$$V_\infty = (V_{BO}^2 - V_p^2)^{1/2} \tag{9}$$

to calculate a resultant planetocentric hyperbolic excess velocity V_∞. By Eqs. 5 and 8 the hyperbolic excess velocity is a function of the parameters: F, I_s, W_0, and t.

Thus for propulsion maneuvers, the parameters V_h, V_a, I_s, and Ft/W_0 which is the total impulse-to-(initial) earth weight ratio have been selected to construct nomographs for analyzing the interrelation of mission and propulsion system requirements. The hyperbolic excess velocities represent the mission parameters—i.e., launch date and transfer

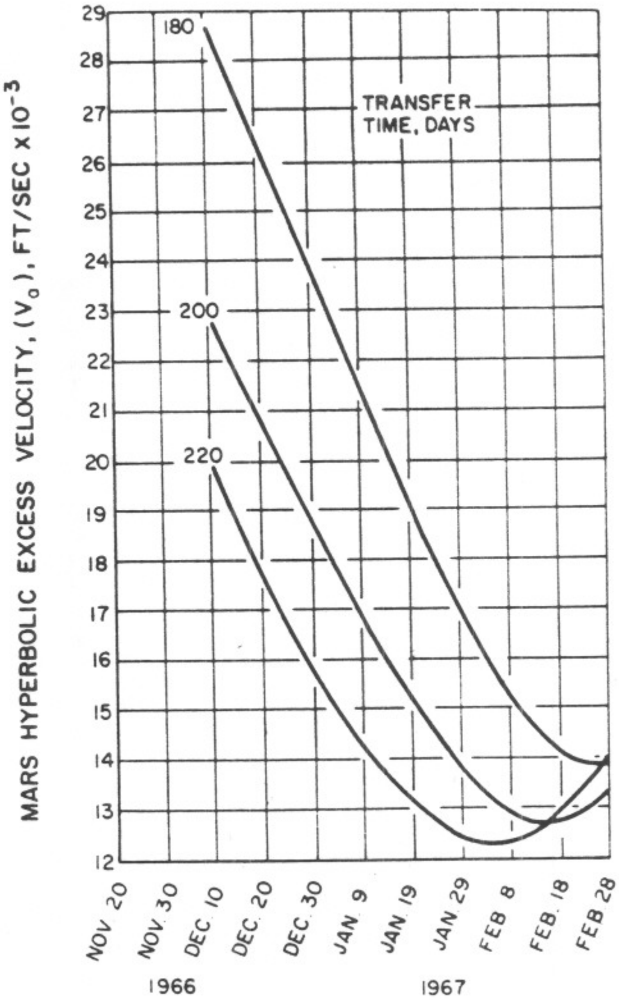

Fig. 3. Earth-Mars planetary mission.

time. The two parameters (I_s, Ft/W_0) are characteristic of the propulsion system.

In the computer simulated trajectories used to generate data for constructing the nomographs, the propulsion phase initiates with the vehicle in a 300-nm planetocentric circular orbit and terminates after the vehicle achieves the planetocentric energy level associated with the required hyperbolic excess velocity (Eq. 9).

For planetary hyperbolic escape maneuvers, this simulation is an accurate representation of the actual vehicle trajectory. However, for planetary hyperbolic capture maneuvers, the simulated trajectory is not identical with the actual trajectory. Although the shapes of an escape trajectory and a capture trajectory are not identical (for a particular planetary hyperbolic excess velocity), the difference in the total impulse-to-weight ratios required to perform the two maneuvers is negligible for hyperbolic excess velocities up to 30,000 ft/sec. This fact has been established by comparison of escape maneuver results with the results of capture maneuvers obtained by the vehicle following a flight path backwards out of the planetocentric orbit with a negative mass flow rate. Thus, the nomographs are equally valid for hyperbolic escape or capture maneuvers.

Nomographs are presented in Figs. 6, 7, and 8, for earth, Venus, and Mars hyperbolic escape or capture maneuvers. These nomographs are based upon planetocentric orbital altitudes of 300 nm and initial thrust-to-mass ratios equivalent to 1.0 thrust-to-weight (planet surface) ratios. Variations in the two parameters, orbital altitude h and initial thrust-to-weight ratio F/W, which change the velocity losses during the propulsion phase, can be accounted for in the analysis method by a slight change in the total impulse-to-weight ratio obtained from the nomograph.

To illustrate the dependency of the total impulse-to-weight ratio upon orbital altitude, an earth orbit departure maneuver has been examined. The altitude of geocentric parking orbits for launching planetary missions is expected to range between 100 to 500 nm.

The 300-nm parking orbit altitude has been selected as representative for analysis of planetary missions. This altitude is sufficiently high to minimize orbital decay and low enough to reduce the radiation problems associated with the inner Van Allen radiation belt that begins at about 500 nm.[4,5]

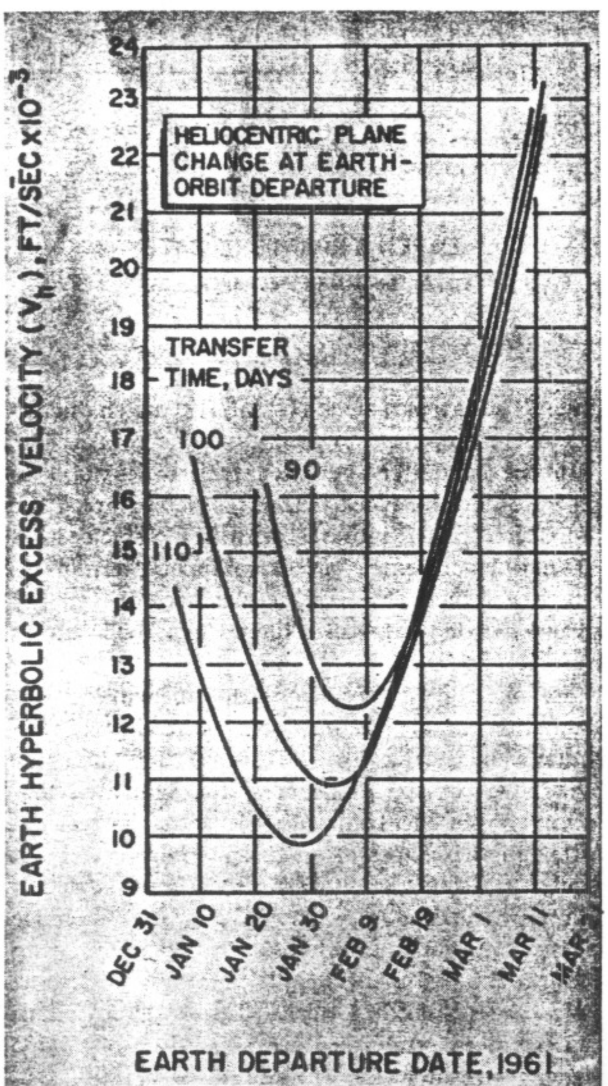

Fig. 4. Earth-Venus mission.

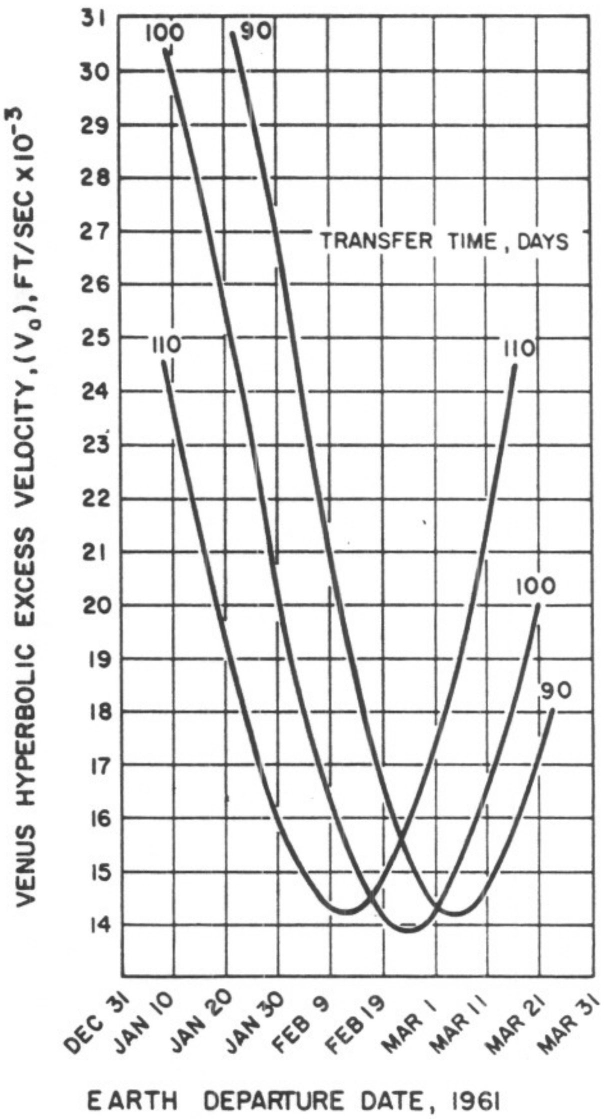

Fig. 5. Earth-Venus mission.

Within the 100- to 500-nm range of initial altitudes for orbit departure, the total impulse-to-weight ratio required for any particular mission parameter V_h or V_a decreases with increasing parking orbit altitude. However, for a representative mission parameter (Fig. 9), the percent deviation about the nominal value is so small, the nomograph can be considered valid for any initial altitude within the range.

The initial F/W ratio has a significant effect upon the total impulse-to-weight ratio obtained from the nomograph. Reducing the initial F/W ratio increases the velocity losses and thereby necessitates a larger total impulse. Since the nomographs (Figs. 6, 7, and 8) are plotted to include the losses for initial thrust-to-weight (planetary surface) ratios of 1.0; a multiplicity factor must be applied to the nomograph impulse-to-weight ratio to obtain a corrected value for other initial F/W ratios. These multiplicity factors can be obtained from Figs. 10–12.

Representative Analyses of Propulsion Requirements

Applicability of the method for obtaining preliminary design information will now be discussed. Three aspects of interplanetary mission analysis will be examined utilizing the nomograph analysis method:

(1) Payload capabilities.
(2) Thrust optimization.
(3) Gross weight prediction.

Payload Capabilities of Interplanetary Space Vehicles

An evaluation of the payload capability of two, typical, propulsion system/vehicle configurations has been performed for a Mars mission. (An example in the Appendix illustrates the nomograph method of analyzing a mission to determine payload). The mission initiates in a 300-nm geocentric orbit. The initial weight W_0 of the space vehicle is the payload weight of booster vehicle configurations performing a 300-nm orbit mission (Table 1). A 200-day transfer, departing the earth orbit on January 15, 1967, is representative of Mars mission parameters τ and T.

For the two-stage space vehicle, cryogenic propellants (liquid oxygen/liquid hydrogen) with a specific impulse I_s of 420 sec have been used in each stage. A representative propellant fraction λp of 0.915 has been used for each stage in arriving at the stage payload weight. Based on results using the nomograph analysis method, Table 2 presents the space vehicle descriptions and payloads placed into a 300-nm Mars orbit by the space vehicles.

Table I. Booster Vehicle Description

Mission: 300-nm Earth Orbit

Vehicle	Booster Vehicle A	Booster Vehicle B
Payload, lb	120,000	350,000
Stage One		
Engine	2 F-1	6 F-1
Propellant	liquid oxygen/RP	liquid oxygen/RP
Gross weight, lb	2,222,000	6,660,000
Sea-level thrust, lb	3,000,000	9,000,000
Propellant weight, lb	1,410,000	4,220,000
Hardware weight, lb	90,000	270,000
Stage Two		
Engine	4 J-2	12 J-2
Propellant	liquid oxygen/ liquid hydrogen	liquid oxygen/ liquid hydrogen
Gross weight, lb	722,000	2,170,000
Altitude thrust, lb	800,000	2,400,000
Propellant weight, lb	555,500	1,670,000
Hardware weight, lb	48,500	146,000

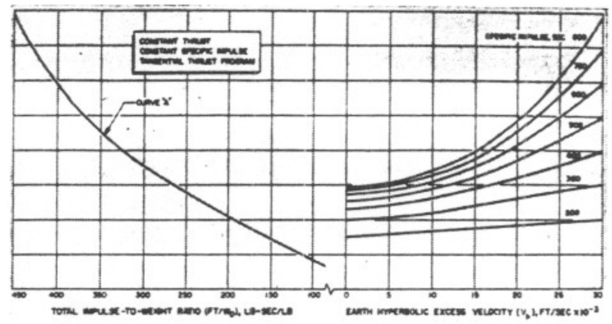

Fig. 6. Earth hyperbolic velocity nomograph.

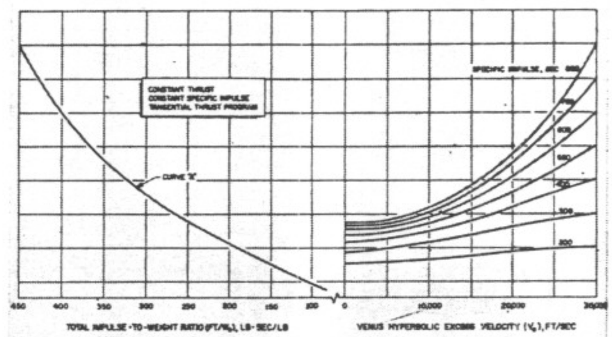

Fig. 7. Venus hyperbolic velocity nomograph.

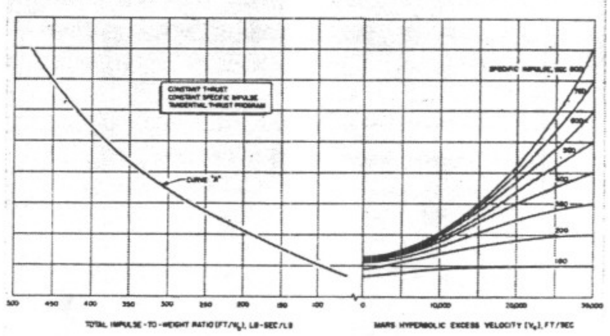

Fig. 8. Mars hyperbolic velocity nomograph.

Thrust Level Optimization

Another application of the nomograph analysis method is the selection of a stage thrust level for a space vehicle to obtain maximum payload from a stage of determined size. Optimization of the thrust level is a tradeoff between reduction of the stage propulsion system weight by lowering the thrust level, and an increase in the velocity losses which result from a decrease in thrust level. Thus, there

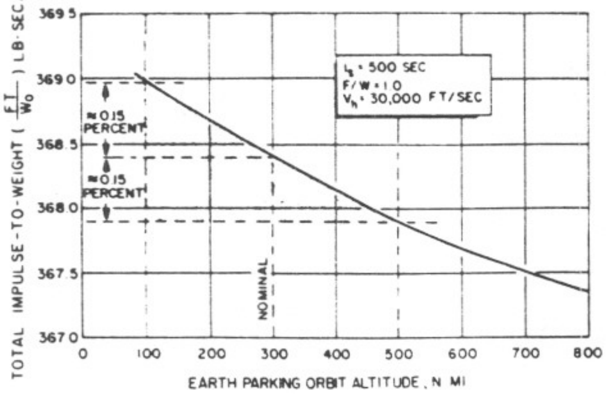

Fig. 9. Variation of total impulse-to-weight ratio with parking orbit altitude.

is a unique thrust level which gives maximum payload.

Once the mission parameters T and τ have been selected, the hyperbolic excess velocity for a stage determined, and the total impulse-to-weight ratio established, curves can be generated to show the relationship between payload, gross weight, and thrust level of the stage. A representative space vehicle thrust level optimization is illustrated in Figs. 13 and 14 for the two stages of a space vehicle performing a Mars mission. The optimum thrust level, or the effect of employing a nonoptimum thrust level can be determined.

Table 2. Space Vehicle Description

Mission: Transfer from 300-nm Earth Orbit to 300-nm Mars Orbit
Departure Date: Jan. 15, 1967

Vehicle	Space Vehicle A	Space Vehicle B
Payload, lb	16,500	49,500
Stage One		
Gross weight, lb	120,000	350,000
Thrust,* lb	60,000	175,000
Propellant weight, lb	70,000	205,000
Hardware weight, lb	6,500	19,000
Propellant	liquid oxygen/ liquid hydrogen	liquid oxygen/ liquid hydrogen
Specific impulse, sec	420	420
Stage Two		
Gross weight, lb	43,500	126,000
Thrust,* lb	8,500	25,000
Propellant weight, lb	24,500	70,000
Hardware weight, lb	2,500	6,500
Propellant	liquid oxygen/ liquid hydrogen	liquid oxygen/ liquid hydrogen
Specific impulse, sec	420	420

* Thrust level assumed to give an initial thrust-to-weight (planet surface) ratio of 0.5.

Manned Space Mission Requirements

For manned interplanetary missions, the payload capsule weights[6] have been predicted. A three-man capsule, including radiation shielding for a Mars orbit-and-return mission, required approximately 60,000 to 80,000 lb of payload. With a payload

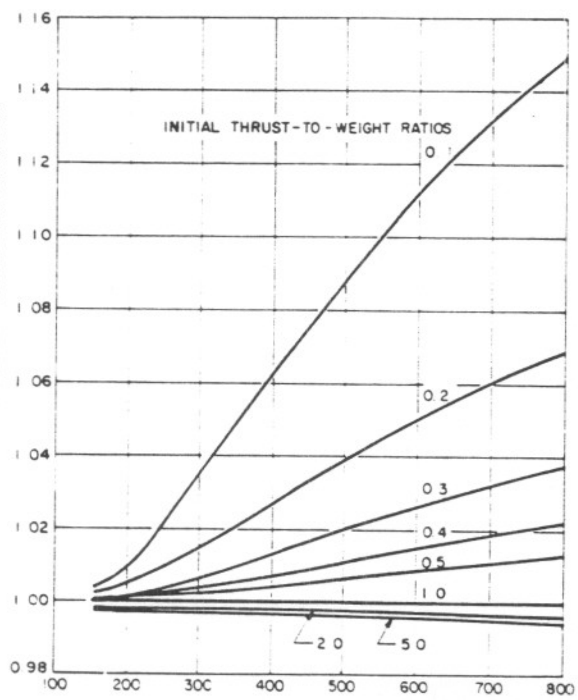

Fig. 10. Total impulse-to-weight correction factor.

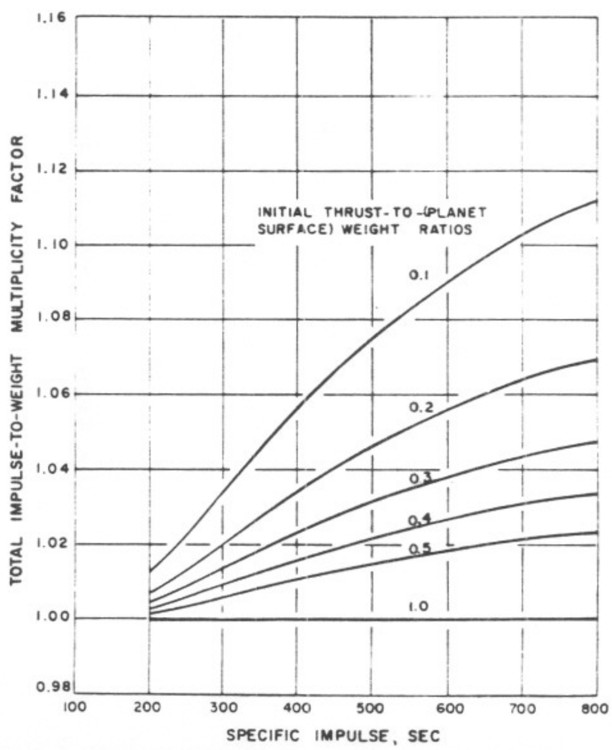

Fig. 11. Venus total impulse-to-weight correction factor.

of 80,000 lb and representative stage design weight criteria, the nomographs have been used to find the approximate initial gross weight of a manned space vehicle. The return from a Mars orbit to an earth orbit will be based on the same velocity requirements as the transfer from an earth orbit to a Mars orbit. In analyzing this space mission, cryogenic propellants have been used for the initial transfer, and noncryogenic (earth-storable) propellants used for the return transfer.[7] The initial gross weight of the space vehicle for a round-trip manned mission between a 300-nm earth orbit and a 300-nm Mars orbit is presented in Table 3. This mission has a 200-day (each way) transfer and an earth orbit departure date of Jan. 15, 1967. It should be noted that a wait time at Mars, of approximately 500 days, has been used for this mission.

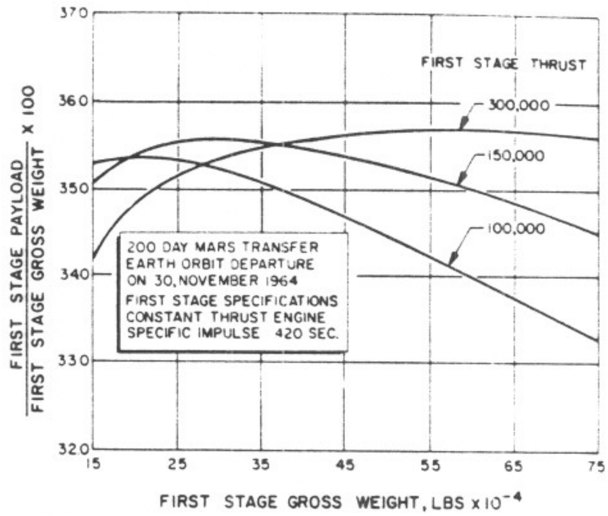

Fig. 13. First-stage thrust level optimization.

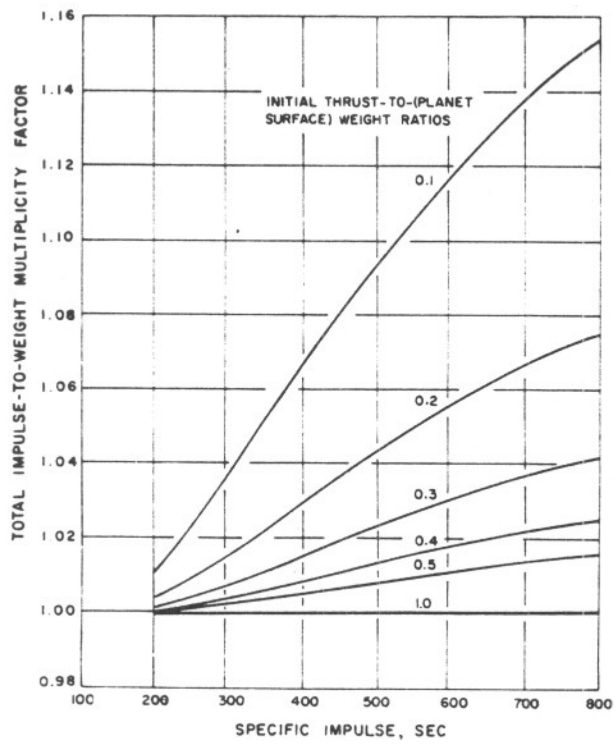

Fig. 12. Mars total impulse-to-weight correction factor.

From Fig. 2, the earth hyperbolic velocity corresponding to the mission parameters T and τ is 10,930 ft/sec. Find this value on the nomograph of Fig. 6. By reading vertically to the point of intersection with the 400-sec specific impulse curve, moving horizontally to an intersection with Curve A, then downward, a value for the total impulse-to-weight ratio equal to 234 lb-sec/lb is obtained. For the assumed parameters, the space vehicle has an initial F/W ratio of 0.2. Thus, the total impulse-to-weight ratio must be changed by the multiplicity factor (1.0272) obtained from Fig. 10. The resultant value of 250.1 is divided by the specific impulse to obtain a factor (Ft/W_0I_s) equal to 0.6253. From Eq. 5, the mass ratio is computed to be 2.669, then

Table 3. Space Mission Performance Comparison

CHEMICAL			
STAGE	PROPELLANT	I_s SEC	PROPELLANT FRACTION *
1	LIQUID OXYGEN/LIQUID HYDROGEN	420	0.915
2	LIQUID OXYGEN/LIQUID HYDROGEN	420	0.915
3	MIXED OXIDES OF NITROGEN/ MONO-METHYL HYDROZINE	310	0.935
4	MIXED OXIDES OF NITROGEN/ MONO-METHYL HYDROZINE	310	0.935

* REPRESENTATIVE FOR PUMP-FED SYSTEMS

Notes:

(1) Three-man Mars orbit mission.
(2) 900-day round trip.
(3) Departure and destination: 300-nm earth orbit.
(4) 80,000-lb-payload capsule.

Appendix

Instructions for an analysis of an earth orbit departure maneuver for a hypothetical space vehicle on a Mars transfer mission are presented. To illustrate the nomograph method, the following mission and first-stage propulsion system parameters will be used:

T = Jan. 15, 1967
τ = 200 days
W_0 = 10,000 lb (earth surface weight)
I_s = 400 sec
F = 2,000 lb
λ_p = 0.90

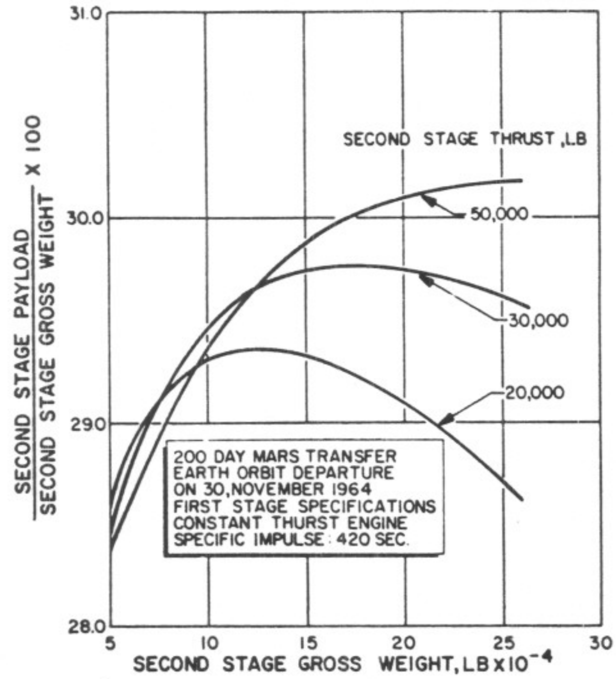

Fig. 14. Second-stage thrust level optimization.

the burnout weight of the vehicle after earth orbit departure is calculated by $W_{BO} = W_0/R$ to be 3;745 lb.

The stage payload weight W_{pl} is calculated in the following manner. First, the weight of the expelled propellants W_p is determined as the difference between the initial weight and the burnout weight

$$W_p = W_0 - W_{BO} \tag{10}$$

Then, with the propellant fraction (λ_p) of a stage

$$\lambda_p = \frac{W_p}{W_p + \text{Inert Weight}} \tag{11}$$

where

$$\text{Inert Weight} = W_0 - W_p - W_{pl} \tag{12}$$

the stage payload weight is calculated from the equation

$$W_{pl} = W_0 - (W_p/\gamma_p) \tag{13}$$

to be 3,050 lb.

For a two-stage vehicle, the payload weight of the first stage is the initial weight of the second stage. Having the initial weight of the second stage, a procedure identical to that described in the foregoing two paragraphs is used to obtain the payload into the Mars orbit.

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